TECHNICAL NOTE

D-126

A HIGH-PERFORMANCE 250-POUND-THRUST ROCKET ENGINE UTILIZING COAXIAL-FLOW INJECTION OF JP-4 FUEL AND LIQUID OXYGEN

By Samuel Stein

Lewis Research Center
Cleveland, Ohio

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A HIGH-PERFORMANCE 250-POUND-THRUST ROCKET ENGINE UTILIZING COAXIAL-FLOW INJECTION OF JP-4 FUEL AND LIQUID OXYGEN

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SUMMARY

Small, compact rocket engines with a high degree of reliability for continual operation were developed for wind-tunnel missile and space-vehicle-booster models. These engines have design features permitting inexpensive and easy fabrication of injectors and combustors for acceleration of research programs.

These engines had high performance that was principally due to a novel means of atomizing liquid fuel by minimizing the mass mean drop size through the mechanism of thin annular streams surrounding the oxidant.

INTRODUCTION

The research conducted in supersonic wind tunnels on rocket exhaust and related external-flow problems of missiles and space-vehicle boosters has established a need for small-scale rocket engines. These engines, preferably in the thrust-level range of 250 to 1000 pounds, must have a high degree of reliability for long burning runs and must be capable of reignition repeatedly without wind-tunnel shutdown.

In the development of the 250-pound-thrust engine of this report, several design criteria were specified. The first requirement was to produce a combustor that could be inexpensively and easily fabricated and that would be separate from the injector head so as to allow injector-head interchangeability with other combustors having the same thrust level but varying in nozzle geometry. Further, a high-performance engine was required to eliminate effects of combustion inefficiencies on base-temperature measurements.

To realize these objectives, a new approach was taken in the design of both components. The injector differed from the usual designs in the dispersion of the jets over the injector face and the relation of the propellants to each other upon injection. The combustor differed from the usual design by the use of a precision-cast combustion chamber. This report describes the design features and performance test of a 250-pound-thrust engine utilizing coaxial-flow injection.

DESIGN CONSIDERATIONS

Rocket engines for missile and space-vehicle models must have external dimensions that permit their clustering and gimbaling in a spaced relation determined by the thrust scale factor. In general, this requires a short engine having a minimum over-all wall thickness, free of external flanges and piping. Additionally, the injector head should not be of greater diameter than the combustor. Within these confines provision must be made for cooling-water flow to and from the combustor wall, a pressure tap, thermocouple, igniter, liquid oxygen and JP-4 fuel connections, and related jets for the atomization of the propellants.

Combustor Design

Two techniques of combustor fabrication were employed. The first was a conventional spun-nickel combustion chamber with twelve coolant-flow passages formed by welding longitudinal wire ribs to the combustion-chamber wall; the shell was wrapped with AM 350 stainless-steel wire and brazed in a vacuum furnace.

This type of construction is well suited for the production of a small number of engines. The principal disadvantage in manufacture is the requirement for a vacuum furnace for the brazing cycles. Another disadvantage is that there is a limited range of geometry for one-piece spinnings dependent on contraction ratio, throat size, and expansion ratio of the chamber. Beyond this range the chamber is formed by welding several spun sections together. These engines, however, were satisfactory in operation and withstood chamber pressures up to 820 pounds per square inch gage. Cooling-water flow limitations prevented further exploration with higher chamber pressures.

The second type of construction employed a precision casting of the combustion chamber with integral ribs (figs. 1 and 2). The material was 1/2-percent-beryllium copper casting formed by the lost wax process. The outer shell was formed by copper-wire wrap and salt-bath brazing.

The cast engine offers a material having a yield strength of 50,000 psi as compared with 20,000 for nickel, and a thermal conductivity three times that of nickel. The wax pattern is formed by dipping a soft brass mandrel in molten wax and then machining to proper wall thickness on a profiling lathe. Wax ribs and gating are attached, and the pattern is invested. The completed casting is machined externally to obtain the designated rib height and, thus, coolant-passage dimension. The shell is then wrapped with copper wire and silver-brazed in a molten salt bath. The as-cast internal surfaces are not machined other than at the forward end of the chamber for accommodating the injector. These engines are inexpensive; a total of 24 man-hours was expended for production from forming the pattern to the completed motor. Thus, for a research program involving various nozzle configurations, a corresponding change in the soft brass mandrel can produce a series of castings.

The technique for casting 1/2-percent-beryllium copper resulted from an extensive program by the metallurgical section of the Lewis Research Center. The cast type of construction has been used in 250- and 500-pound-thrust engines thus far; further use is expected in view of the high strength of the material, its heat-transfer properties, resistance to erosion, low cost factor, and ease in fabrication.

Injector Design

Normally the propellant injection points are dispersed over the full face area of the injector; however, in order to include instrumentation and to avoid complexity in fabrication, the jets were concentrated in the central part of the injector face over an area roughly 10 percent that of the full injector face, as shown in figure 3.

The following rocket-engine specifications were required:

Combustion length, in.	3
Chamber pressure, lb/sq in.	600
Contraction ratio	5.5
Combustion efficiency, percent	95

The length specified was about one-half that normally considered for the fuel combination of liquid oxygen and JP-4 fuel; this indicated the necessity for achieving efficient combustion of the mixture through the mechanism of providing smaller mass mean drop size than previously used.

Reference 1 indicates that performance characteristics of this nature would require an injector which produces a mass mean drop size for liquid oxygen no greater than 0.0027 inch and for JP-4 fuel no greater than 0.0017 inch. Previous combustor tests (ref. 2) indicated that the mass mean drop radius produced by various size jets is approximately equal to one-tenth of the jet diameter. The drop-size requirements therefore indicate that jet diameters of about 0.027 inch for liquid oxygen and 0.017 inch for JP-4 fuel would be required. Inasmuch as drilling 0.017-inch-diameter holes is a tedious operation and the available space on the injector for jets was limited to about 10 percent of the full face area, it appeared that the contemplated shower-head design could be modified so that the fuel flow would be annular around the liquid-oxygen streams instead of adjacent to them. Although shower-head injectors have been operated with the fuel jets surrounded by annular jets of oxidant, the technique of surrounding the oxidant jets with annular fuel jets is new and, as far as is known, has not been used previously.

The first injector fabricated had 0.028-inch-diameter liquid-oxygen streams issuing from 0.041-inch-outside-diameter hypodermic tubes with fuel injected around each liquid-oxygen stream from an annulus having a flow area equivalent to that of a 0.017-inch-diameter orifice. Performance was good for this injector, but the pressure drop for the fuel proved excessive. The second injector was designed on the basis of using 0.025-inch-diameter fuel streams, which were designated in reference 1 as the maximum allowable at the lower injection velocity accompanying the selected lower pressure drop. The equivalent of this 0.025-inch-diameter stream requirement for the fuel was fulfilled by increasing the outer diameter of the annular fuel stream to 0.049 inch.

Although of the same cross-sectional area as a solid stream, the annular stream can be more readily vaporized than an equivalent solid stream because of its increased exposed surface area and reduced thickness. This, together with the fact that the diameter of the liquid-oxygen stream is less than the allowable maximum, predicts a high combustion efficiency for the rocket engine.

The parts of the injector head (fig. 4) are made of 304 stainless steel and are vacuum-furnace-brazed into the assembly shown in figure 3. Contained in the head are 45 stainless-steel hypodermic tubes, 0.041-inch outside diameter by 0.0065-inch wall thickness, forming the liquid-oxygen passages. These tubes are positioned in a corresponding number of holes of 0.049-inch diameter. The location of the holes is within ordinary machine-shop tolerances determined from a drill jig. The alinement of the tubes in the holes is not critical inasmuch as combustion tests indicated no noticeable difference as to whether the tubes are concentric with the hole or displaced so as to form a crescent-shaped path for the fuel.

The inner member of each jet stream is that of liquid oxygen gasified as it leaves the hypodermic tubing. This sudden expansion atomizes the liquid fuel as it is introduced from the 0.004-inch-thick annular stream around the oxidant. The close spacing of the coaxial jets assures thorough mixing and equalized dispersion of any drops of larger than normal size. This thorough mixing near the center of the combustion chamber minimizes the possibility of burnout of the combustion-chamber wall, as might occur with a misdirected or uncombined stream of oxygen.

The propellant flow through the injector is shown in figure 5 as a function of the pressure drop across the injector. Calibration was made by water flow and was corrected for density. This calibration was in substantial agreement with actual flow measurements made during combustion tests.

The igniter shown in cross section in figure 2 was brazed into the injector head. This device uses a spark plug or glow plug for ignition of the fuel combination of propane and gaseous oxygen. The flame is supersonic as it leaves the nozzle and enters the rocket-engine combustion chamber. Provision is made for a thermocouple to indicate the igniter flame temperature prior to actuating the main propellant flow.

The injector assembly is relatively easy to fabricate. Approximately 50 man-hours were required after necessary tooling was made.

EXPERIMENTAL TEST APPARATUS

The motors were tested on small-scale test facilities at sea-level conditions. The thrust rig shown in figure 6 utilizes a calibrated strain-gage element. The effect of the pressurized piping was included in the calibration of the thrust stand.

JP-4 fuel and liquid oxygen were supplied from pressurized systems. Flows were measured on rotary-vane-type flowmeters, carefully calibrated in view of the relatively low flow rates. The combustion-chamber pressure was determined by a strain-gage pressure transducer.

Runs averaging 2 to 3 minutes in duration were made to assure steady-state conditions with special regard to the weight flow of the liquid oxygen in view of its density-temperature relation. Erosion resistance was evaluated on the basis of longer runs, up to 15 minutes duration.

The over-all accuracy of the data is within ±2 percent.

RESULTS AND DISCUSSION

The experimentally determined specific-impulse curve is compared in figure 7 with the theoretical curves at the same expansion ratio and sea-level condition. Evaluation of the engine shows that its performance is comparable with that of full-scale engines, the specific impulse being approximately equal to that for frozen composition and within about 3 percent of the equilibrium values. The high performance in conjunction with a characteristic length of 14 inches as compared with about 40 inches for full-scale engines indicates that coaxial injection of this pattern provides good atomization and mixing of the propellants.

Independent measurements of characteristic velocity were in good agreement with the specific impulse, in that the characteristic velocity was within a few percent of the theoretical value for equilibrium composition.

Tests on similar engines with variations in the combustion-chamber length revealed that there was no appreciable increase in performance at 600 pounds per square inch chamber pressure for engines having a characteristic length greater than 14. However, an engine having a 2-inch-long combustion chamber with a characteristic length of 9 showed a 6 percent decrease in specific impulse. An engine with a chamber length of $1\frac{1}{4}$ inches and a characteristic length of 4 was erratic in operation, the flame separating from the nozzle aperiodically.

Ignition of the main propellants was relatively smooth. However, erratic shutdowns indicated the necessity for improved purging. The igniter functioned well from sea level to an altitude of about 65,000 feet, as determined in a separate wind-tunnel test. Both the glow plugs and spark plugs withstood the vibration during combustion and enabled repeated starts.

The fatigue life of the engines was not fully explored; however, it is known that elongation during combustion is within the elastic range. Measurements made after 24 firings did not reveal any change in the original measurements. There was no evidence of erosion, and throat measurements agreed with the original dimensions. Although the liquid-oxygen tank size limited the length of combustion time to 15 minutes for single runs, it is believed that susceptibility to erosion would be fully indicated within this period.

SUMMARY OF RESULTS

Small size, high-performance rocket engines have been developed. These engines are suitable for use in wind-tunnel models of missiles or space-vehicle boosters and offer a compact unit for cluster arrangement and directional control. An ignition system was developed for starting the engines at the altitudes simulated in wind tunnels and for repeated operation without tunnel shutdown. The engines have the combustion performance expected of full-scale engines but have disproportionately smaller chamber volumes and length.

The high performance of the engine was principally due to the novel method of injecting liquid fuel in annular jets around the oxidant. The reliability of the engines was principally due to the centralized location of the propellant jets to assure good mixing and minimize the possibility of chamber burnout from uncombined or misdirected streams of oxidant. The method of fabricating the combustor from a precision casting has proven less costly and faster than conventional types of fabrication and has minimized the time required for delivery of this research tool.

Lewis Research Center
National Aeronautics and Space Administration
Cleveland, Ohio, July 17, 1959

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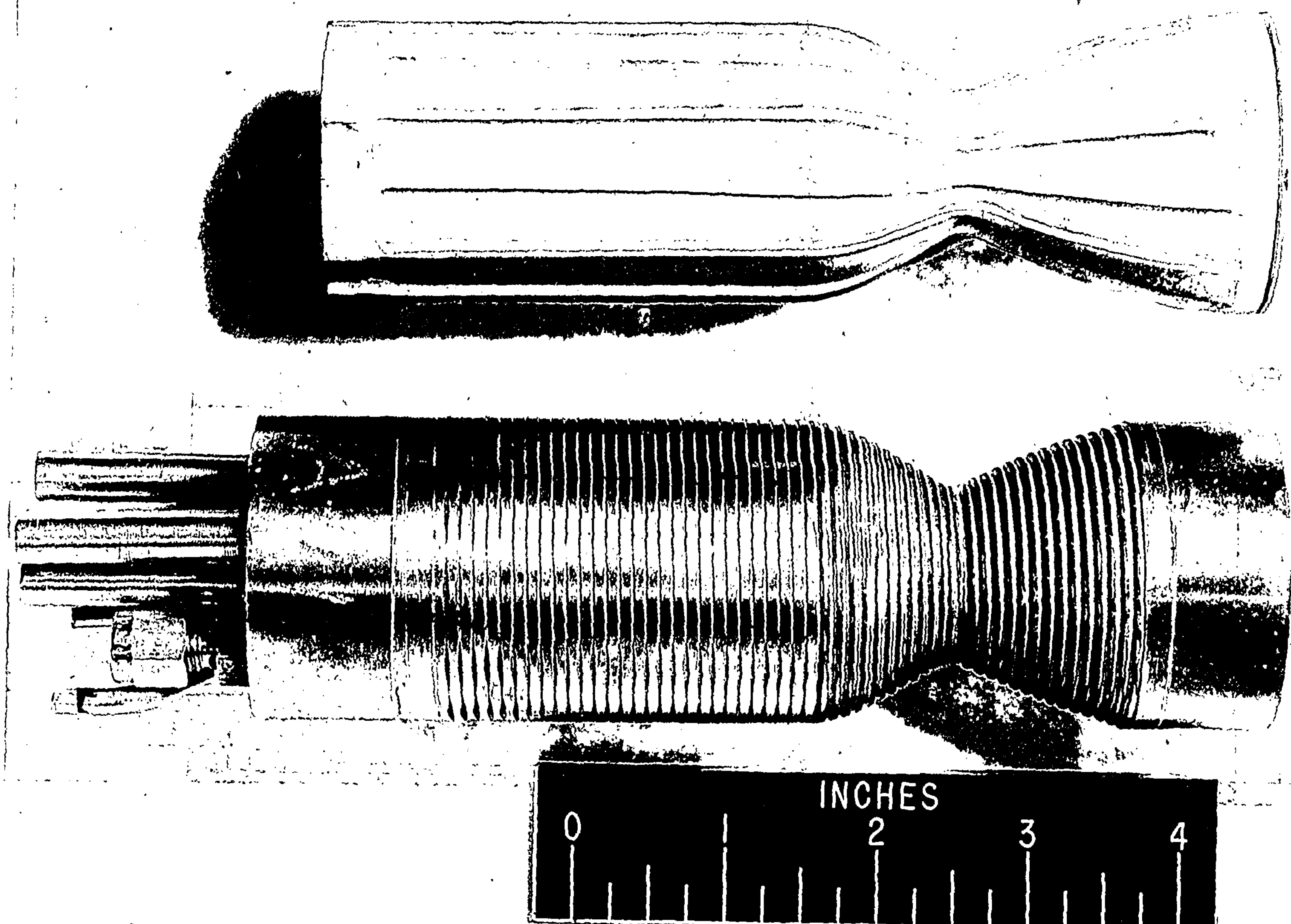

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Figure 1. - Combustor-chamber casting and completed engine assembly.

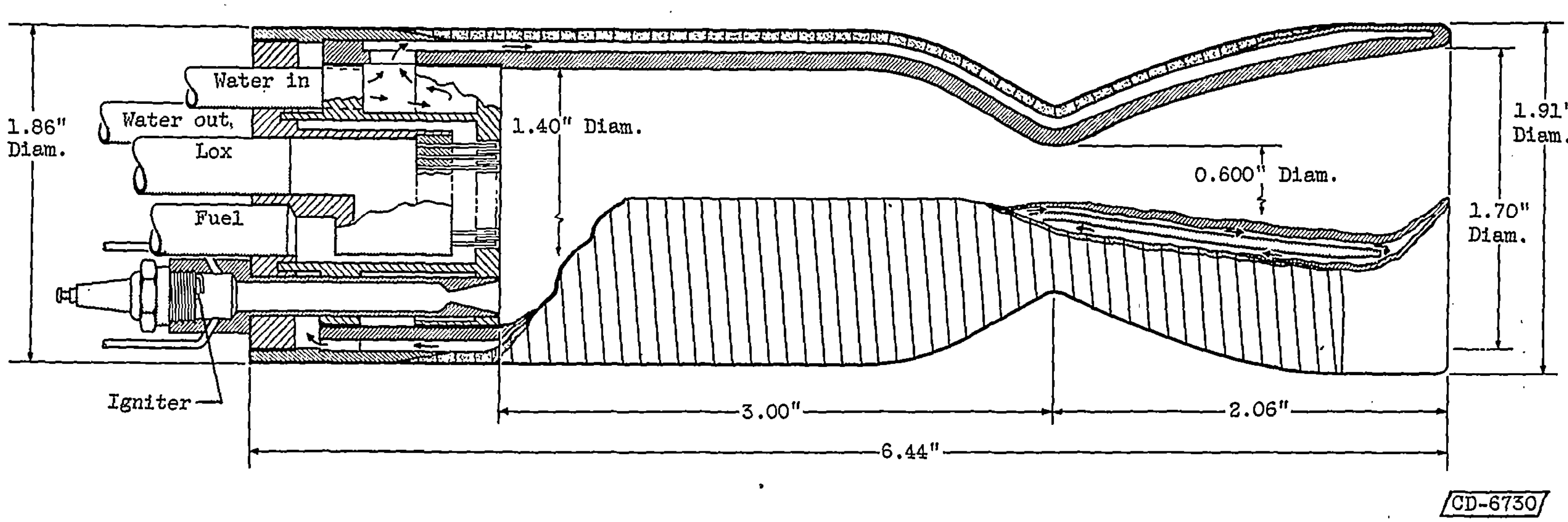

Figure 2. - Cross section of 250-pound-thrust rocket engine. Design combustion-chamber pressure, 600 pounds per square inch; nozzle exit to throat area ratio, 8; combustion-chamber volume, 4.0 cubic inches; throat area, 0.283 square inch; characteristic length, 14 inches.

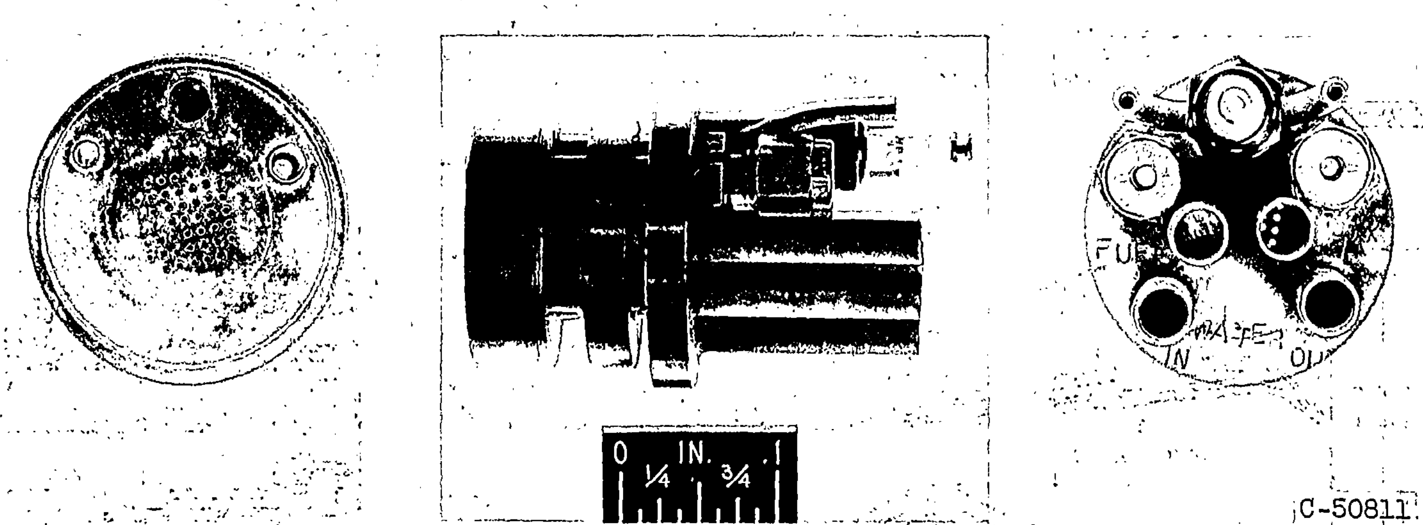

Figure 3. - Injector assembly.

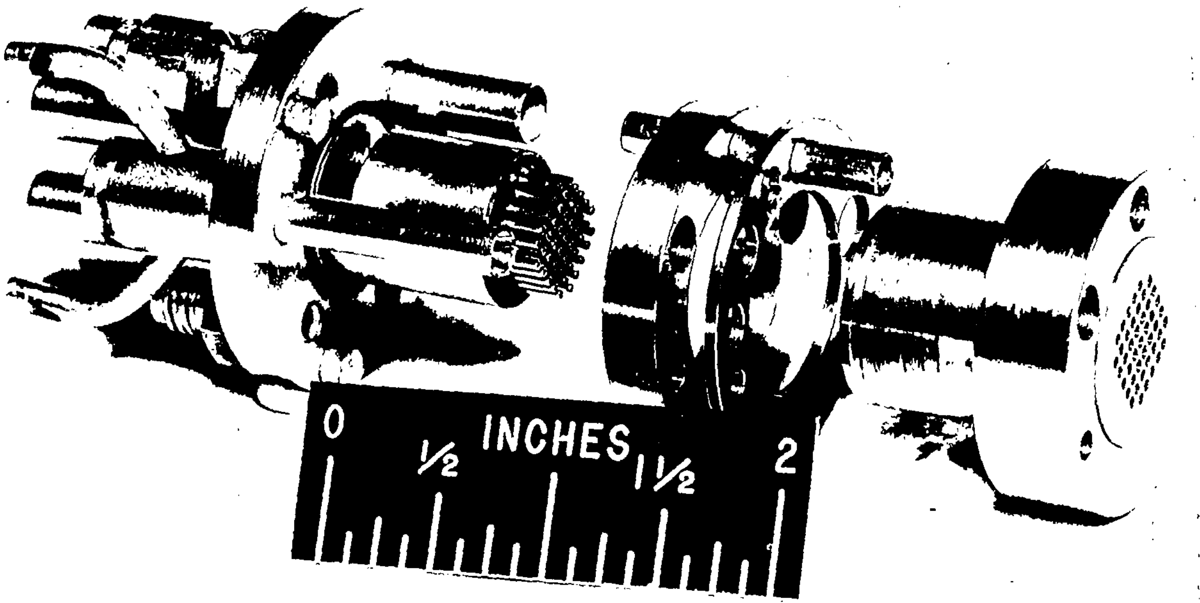

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Figure 4. - Injector parts.

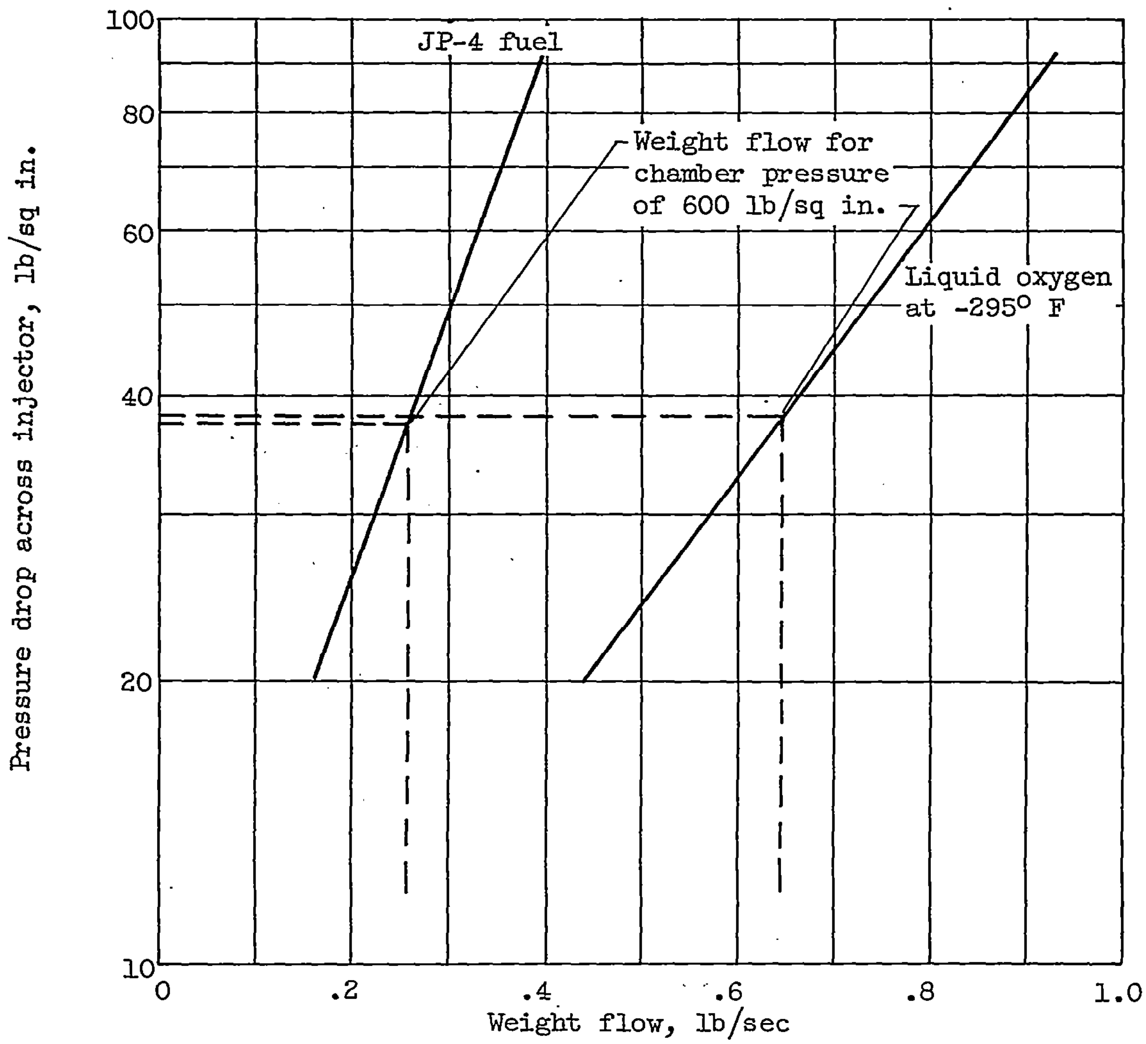

Figure 5. - Flow calibration of injector. Oxidant-fuel ratio, 2.37.

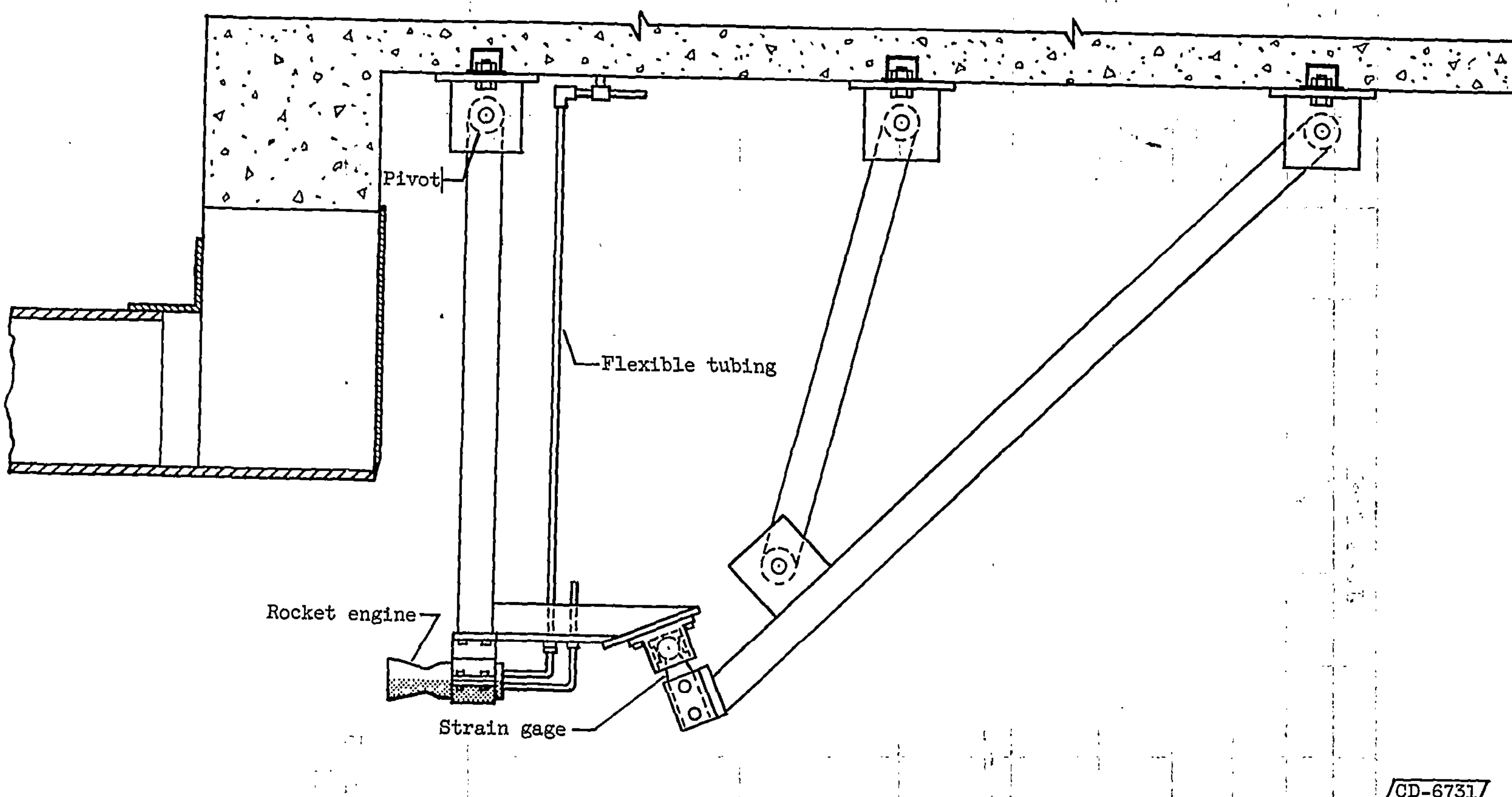

Figure 6. - Test installation, plan view.

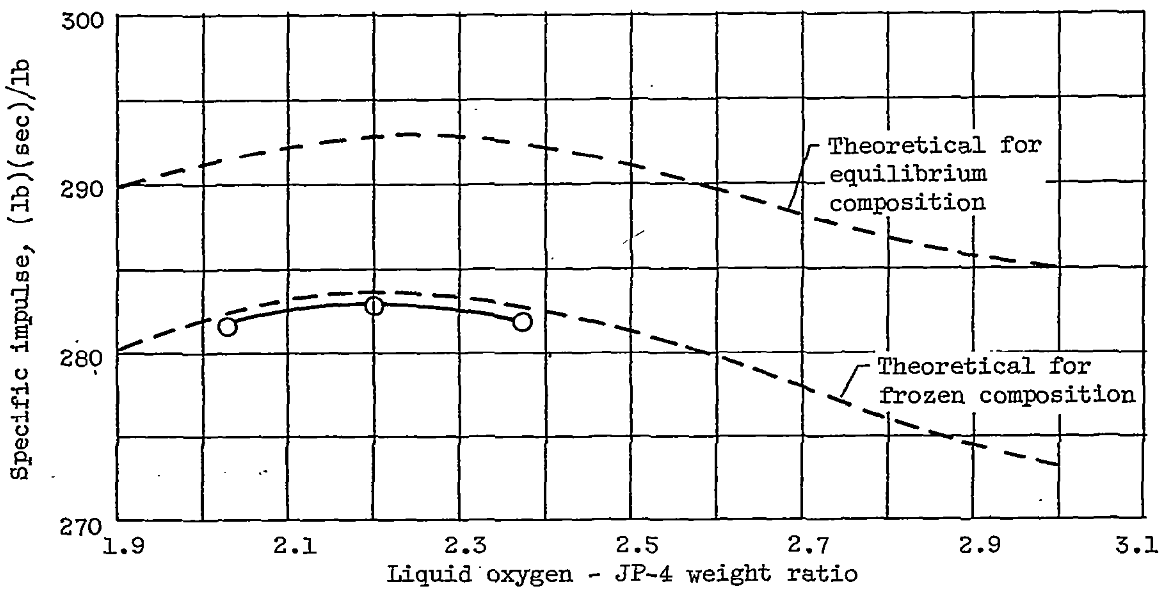

Figure 7. - Specific impulse at various oxidant-fuel weight ratios. Combustion-chamber pressure, 600 pounds per square inch; Bellmouth nozzle expansion ratio, 8:1; sea-level conditions. Theoretical curves obtained from references 3 and 4 for same conditions.

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